(8) The effect, on the net takeoff flight path and on the enroute gradient of climb/descent with one engine inoperative, of 50 percent of the headwind component and 150 percent of the tailwind component;

(9) Overweight landing performance information (determined by extrapolation and computed for the range of weights between the maximum landing and maximum takeoff weights) as follows—

(i) The maximum weight for each airport altitude and ambient temperature at which the airplane complies with the climb requirements of § 23.63(d)(2); and

(ii) The landing distance determined under § 23.75 for each airport altitude and standard temperature.

(10) The relationship between IAS and CAS determined in accordance with § 23.1323 (b) and (c).

(11) The altimeter system calibration required by § 23.1325(e).

[Doc. No. 27807, 61 FR 5194, Feb. 9, 1996]

§ 23.1589 Loading information.

The following loading information must be furnished:

(a) The weight and location of each item of equipment that can be easily removed, relocated, or replaced and that is installed when the airplane was weighed under the requirement of § 23.25.

(b) Appropriate loading instructions for each possible loading condition between the maximum and minimum weights established under § 23.25, to facilitate the center of gravity remaining within the limits established under § 23.23.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23–45, 58 FR 42167, Aug. 6, 1993; Amdt. 23–50, 61 FR 5195, Feb. 9, 1996]

APPENDIX A TO PART 23—SIMPLIFIED DESIGN LOAD CRITERIA

A23.1 *General.*

(a) The design load criteria in this appendix are an approved equivalent of those in §§ 23.321 through 23.459 of this subchapter for an airplane having a maximum weight of 6,000 pounds or less and the following configuration:

(1) A single engine excluding turbine powerplants;

(2) A main wing located closer to the airplane's center of gravity than to the aft, fuselage-mounted, empennage;

(3) A main wing that contains a quarter-chord sweep angle of not more than 15 degrees fore or aft;

(4) A main wing that is equipped with trailing-edge controls (ailerons or flaps, or both);

(5) A main wing aspect ratio not greater than 7;

(6) A horizontal tail aspect ratio not greater than 4;

(7) A horizontal tail volume coefficient not less than 0.34;

(8) A vertical tail aspect ratio not greater than 2;

(9) A vertical tail platform area not greater than 10 percent of the wing platform area; and

(10) Symmetrical airfoils must be used in both the horizontal and vertical tail designs.

(b) Appendix A criteria may not be used on any airplane configuration that contains any of the following design features:

(1) Canard, tandem-wing, close-coupled, or tailless arrangements of the lifting surfaces;

(2) Biplane or multiplane wing arrangements;

(3) T-tail, V-tail, or cruciform-tail (+) arrangements;

(4) Highly-swept wing platform (more than 15-degrees of sweep at the quarter-chord), delta planforms, or slatted lifting surfaces; or

(5) Winglets or other wing tip devices, or outboard fins.

A23.3 *Special symbols.*

n_1=Airplane Positive Maneuvering Limit Load Factor.

n_2=Airplane Negative Maneuvering Limit Load Factor.

n_3=Airplane Positive Gust Limit Load Factor at V_C.

n_4=Airplane Negative Gust Limit Load Factor at V_C.

n_{flap}=Airplane Positive Limit Load Factor With Flaps Fully Extended at V_F.

$$* V_{F\ min} = \text{Minimum Design Flap Speed} = 11.0\sqrt{n_1 W/S}\ [\text{kts}]$$

$$* V_{A\ min} = \text{Minimum Design Maneuvering Speed} = 15.0\sqrt{n_1 W/S}\ [\text{kts}]$$

$$* V_{C\ min} = \text{Minimum Design Cruising Speed} = 17.0\sqrt{n_1 W/S}\ [\text{kts}]$$

$$* V_{D\ min} = \text{Minimum Design Dive Speed} = 24.0\sqrt{n_1 W/S}\ [\text{kts}]$$

A23.5 *Certification in more than one category.*

The criteria in this appendix may be used for certification in the normal, utility, and acrobatic categories, or in any combination of these categories. If certification in more than one category is desired, the design category weights must be selected to make the term $n_1 W$ constant for all categories or greater for one desired category than for others. The wings and control surfaces (including wing flaps and tabs) need only be investigated for the maximum value of $n_1 W$, or for the category corresponding to the maximum design weight, where $n_1 W$ is constant. If the acrobatic category is selected, a special unsymmetrical flight load investigation in accordance with paragraphs A23.9(c)(2) and A23.11(c)(2) of this appendix must be completed. The wing, wing carrythrough, and the horizontal tail structures must be checked for this condition. The basic fuselage structure need only be investigated for the highest load factor design category selected. The local supporting structure for dead weight items need only be designed for the highest load factor imposed when the particular items are installed in the airplane. The engine mount, however, must be designed for a higher side load factor, if certification in the acrobatic category is desired, than that required for certification in the normal and utility categories. When designing for landing loads, the landing gear and the airplane as a whole need only be investigated for the category corresponding to the maximum design weight. These simplifications apply to single-engine aircraft of conventional types for which experience is available, and the Administrator may require additional investigations for aircraft with unusual design features.

A23.7 *Flight loads.*

(a) Each flight load may be considered independent of altitude and, except for the local supporting structure for dead weight items, only the maximum design weight conditions must be investigated.

(b) Table 1 and figures 3 and 4 of this appendix must be used to determine values of n_1, n_2, n_3, and n_4, corresponding to the maximum design weights in the desired categories.

(c) Figures 1 and 2 of this appendix must be used to determine values of n_3 and n_4 corresponding to the minimum flying weights in the desired categories, and, if these load factors are greater than the load factors at the design weight, the supporting structure for dead weight items must be substantiated for the resulting higher load factors.

(d) Each specified wing and tail loading is independent of the center of gravity range. The applicant, however, must select a c.g. range, and the basic fuselage structure must be investigated for the most adverse dead weight loading conditions for the c.g. range selected.

(e) The following loads and loading conditions are the minimums for which strength must be provided in the structure:

(1) *Airplane equilibrium.* The aerodynamic wing loads may be considered to act normal to the relative wind, and to have a magnitude of 1.05 times the airplane normal loads (as determined from paragraphs A23.9 (b) and (c) of this appendix) for the positive flight conditions and a magnitude equal to the airplane normal loads for the negative conditions. Each chordwise and normal component of this wing load must be considered.

(2) *Minimum design airspeeds.* The minimum design airspeeds may be chosen by the applicant except that they may not be less than the minimum speeds found by using figure 3 of this appendix. In addition, V_{Cmin} need not exceed values of 0.9 V_H actually obtained at sea level for the lowest design weight category for which certification is desired. In computing these minimum design airspeeds, n_1 may not be less than 3.8.

(3) *Flight load factor.* The limit flight load factors specified in Table 1 of this appendix represent the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the airplane) to the weight of the airplane. A positive flight load factor is an aerodynamic force acting upward, with respect to the airplane.

A23.9 *Flight conditions.*

(a) *General.* Each design condition in paragraphs (b) and (c) of this section must be used to assure sufficient strength for each condition of speed and load factor on or within the boundary of a $V-n$ diagram for the airplane similar to the diagram in figure 4 of this appendix. This diagram must also be used to determine the airplane structural operating limitations as specified in §§23.1501(c) through 23.1513 and §23.1519.

(b) *Symmetrical flight conditions.* The airplane must be designed for symmetrical flight conditions as follows:

(1) The airplane must be designed for at least the four basic flight conditions, ''A'', ''D'', ''E'', and ''G'' as noted on the flight envelope of figure 4 of this appendix. In addition, the following requirements apply:

(i) The design limit flight load factors corresponding to conditions ''D'' and ''E'' of figure 4 must be at least as great as those specified in Table 1 and figure 4 of this appendix, and the design speed for these conditions must be at least equal to the value of V_D found from figure 3 of this appendix.

(ii) For conditions ''A'' and ''G'' of figure 4, the load factors must correspond to those specified in Table 1 of this appendix, and the design speeds must be computed using these load factors with the maximum static lift coefficient C_{NA} determined by the applicant. However, in the absence of more precise computations, these latter conditions may be based on a value of C_{NA}=±1.35 and the design speed for condition ''A'' may be less than $V_{Amin.}$

(iii) Conditions ''C'' and ''F'' of figure 4 need only be investigated when n_3 W/S or n_4 W/S are greater than n_1 W/S or n_2 W/S of this appendix, respectively.

(2) If flaps or other high lift devices intended for use at the relatively low airspeed of approach, landing, and takeoff, are installed, the airplane must be designed for the two flight conditions corresponding to the values of limit flap-down factors specified in Table 1 of this appendix with the flaps fully extended at not less than the design flap speed V_{Fmin} from figure 3 of this appendix.

(c) *Unsymmetrical flight conditions.* Each affected structure must be designed for unsymmetrical loadings as follows:

(1) The aft fuselage-to-wing attachment must be designed for the critical vertical surface load determined in accordance with paragraph SA23.11(c)(1) and (2) of this appendix.

(2) The wing and wing carry-through structures must be designed for 100 percent of condition ''A'' loading on one side of the plane of symmetry and 70 percent on the opposite side for certification in the normal and utility categories, or 60 percent on the opposite side for certification in the acrobatic category.

(3) The wing and wing carry-through structures must be designed for the loads resulting from a combination of 75 percent of the positive maneuvering wing loading on both sides of the plane of symmetry and the maximum wing torsion resulting from aileron displacement. The effect of aileron displacement on wing torsion at V_C or V_A using the basic airfoil moment coefficient modified over the aileron portion of the span, must be computed as follows:

(i) *Cm*=*Cm* +0.01δμ (up aileron side) wing basic airfoil.

(ii) *Cm*=*Cm* −0.01δμ(down aileron side) wing basic airfoil, where δμ is the up aileron deflection and δ d is the down aileron deflection.

(4) Δ critical, which is the sum of δμ+δ d must be computed as follows:

(i) Compute Δα and ΔB from the formulas:

$$\Delta_a = \frac{V_A}{V_C} \times \Delta_p \quad \text{and}$$

$$\Delta_b = 0.5 \frac{V_A}{V_D} \times \Delta_p$$

Where ΔP=the maximum total deflection (sum of both aileron deflections) at V_A with $V_{A,}$ $V_{C,}$ and V_D described in subparagraph (2) of §23.7(e) of this appendix.

(ii) Compute *K* from the formula:

$$K = \frac{\left(C_m - 0.01\delta_b\right)V_{D^2}}{\left(C_m - 0.01\delta_a\right)V_{C^2}}$$

where δα is the down aileron deflection corresponding to Δα· and δb is the down aileron deflection corresponding to Δ b as computed in step (i).

(iii) If *K* is less than 1.0, Δα is Δ critical and must be used to determine δU and δ*d*. In this case, V_C is the critical speed which must be used in computing the wing torsion loads over the aileron span.

(iv) If *K* is equal to or greater than 1.0, ΔB is Δ critical and must be used to determine δU and δD. In this case, V_d is the critical speed which must be used in computing the wing torsion loads over the aileron span.

(d) *Supplementary conditions; rear lift truss; engine torque; side load on engine mount.* Each of the following supplementary conditions must be investigated:

(1) In designing the rear lift truss, the special condition specified in §23.369 may be investigated instead of condition ''G'' of figure 4 of this appendix. If this is done, and if certification in more than one category is desired, the value of *W/S* used in the formula appearing in §23.369 must be that for the category corresponding to the maximum gross weight.

(2) Each engine mount and its supporting structures must be designed for the maximum limit torque corresponding to METO power and propeller speed acting simultaneously with the limit loads resulting from the maximum positive maneuvering flight load factor n_1. The limit torque must be obtained by multiplying the mean torque by a factor of 1.33 for engines with five or more cylinders. For 4, 3, and 2 cylinder engines, the factor must be 2, 3, and 4, respectively.

(3) Each engine mount and its supporting structure must be designed for the loads resulting from a lateral limit load factor of not less than 1.47 for the normal and utility categories, or 2.0 for the acrobatic category.

A23.11 *Control surface loads.*

(a) *General.* Each control surface load must be determined using the criteria of paragraph (b) of this section and must lie within the simplified loadings of paragraph (c) of this section.

(b) *Limit pilot forces.* In each control surface loading condition described in paragraphs (c) through (e) of this section, the airloads on the movable surfaces and the corresponding deflections need not exceed those which could be obtained in flight by employing the maximum limit pilot forces specified in the table in §23.397(b). If the surface loads are limited by these maximum limit pilot forces, the tabs must either be considered to be deflected to their maximum travel in the direction which would assist the pilot or the deflection must correspond to the maximum degree of "out of trim" expected at the speed for the condition under consideration. The tab load, however, need not exceed the value specified in Table 2 of this appendix.

(c) *Surface loading conditions.* Each surface loading condition must be investigated as follows:

(1) Simplified limit surface loadings for the horizontal tail, vertical tail, aileron, wing flaps, and trim tabs are specified in figures 5 and 6 of this appendix.

(i) The distribution of load along the span of the surface, irrespective of the chordwise load distribution, must be assumed proportional to the total chord, except on horn balance surfaces.

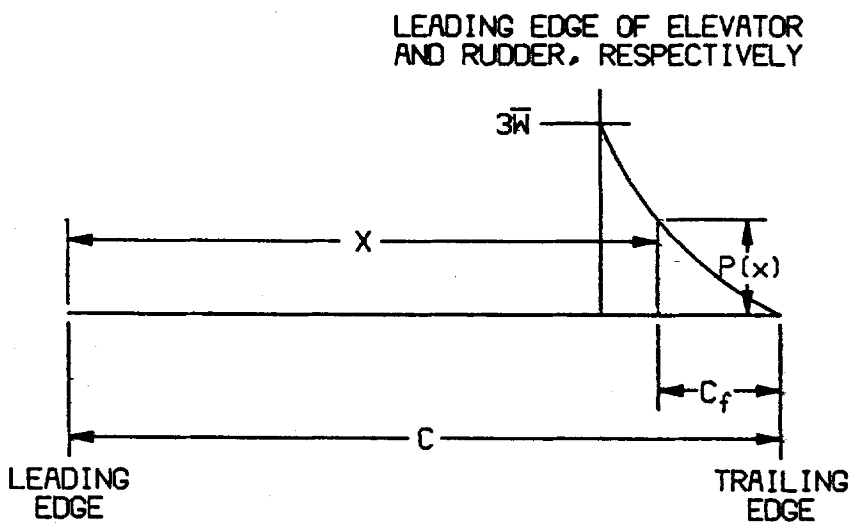
(ii) The load on the stabilizer and elevator, and the load on fin and rudder, must be distributed chordwise as shown in figure 7 of this appendix.

(iii) In order to ensure adequate torsional strength and to account for maneuvers and gusts, the most severe loads must be considered in association with every center of pressure position between the leading edge and the half chord of the mean chord of the surface (stabilizer and elevator, or fin and rudder).

(iv) To ensure adequate strength under high leading edge loads, the most severe stabilizer and fin loads must be further considered as being increased by 50 percent over the leading 10 percent of the chord with the loads aft of this appropriately decreased to retain the same total load.

(v) The most severe elevator and rudder loads should be further considered as being distributed parabolically from three times the mean loading of the surface (stabilizer and elevator, or fin and rudder) at the leading edge of the elevator and rudder, respectively, to zero at the trailing edge according to the equation:

$$P(x) = 3(\overline{w})\frac{(c-x)^2}{{c_f}^2}$$



Where—

P(x)=local pressure at the chordwise stations x,

c=chord length of the tail surface,

c_f=chord length of the elevator and rudder respectively, and

w̌≤=average surface loading as specified in Figure A5.

(vi) The chordwise loading distribution for ailerons, wing flaps, and trim tabs are specified in Table 2 of this appendix.

(2) If certification in the acrobatic category is desired, the horizontal tail must be investigated for an unsymmetrical load of 100 percent *w* on one side of the airplane centerline and 50 percent on the other side of the airplane centerline.

(d) *Outboard fins.* Outboard fins must meet the requirements of § 23.445.

(e) *Special devices.* Special devices must meet the requirements of § 23.459.

A23.13 *Control system loads.*

(a) *Primary flight controls and systems.* Each primary flight control and system must be designed as follows:

(1) The flight control system and its supporting structure must be designed for loads corresponding to 125 percent of the computed hinge moments of the movable control surface in the conditions prescribed in A23.11 of this appendix. In addition—

(i) The system limit loads need not exceed those that could be produced by the pilot and automatic devices operating the controls; and

(ii) The design must provide a rugged system for service use, including jamming, ground gusts, taxiing downwind, control inertia, and friction.

(2) Acceptable maximum and minimum limit pilot forces for elevator, aileron, and rudder controls are shown in the table in § 23.397(b). These pilots loads must be assumed to act at the appropriate control grips or pads as they would under flight conditions, and to be reacted at the attachments of the control system to the control surface horn.

(b) *Dual controls.* If there are dual controls, the systems must be designed for pilots operating in opposition, using individual pilot loads equal to 75 percent of those obtained in accordance with paragraph (a) of this section, except that individual pilot loads may not be less than the minimum limit pilot forces shown in the table in § 23.397(b).

(c) *Ground gust conditions.* Ground gust conditions must meet the requirements of § 23.415.

(d) *Secondary controls and systems.* Secondary controls and systems must meet the requirements of § 23.405.

TABLE 1—LIMIT FLIGHT LOAD FACTORS

[Limit flight load factors]

Flight load factors	Normal category	Utility category	Acrobatic category
Flaps up:			
n_1	3.8	4.4	6.0
n_2	$-0.5\ n_1$	....	....
n_3	(1)	....	....
n_4	(2)	....	....
Flaps down:			
n flap	0.5 n_1	....	....
n flap	[3] Zero	....	....

[1] Find n_3 from Fig. 1

[2] Find n_4 from Fig. 2

[3] Vertical wing load may be assumed equal to zero and only the flap part of the wing need be checked for this condition.

Table 2 - Average limit control surface loading

AVERAGE LIMIT CONTROL SURFACE LOADING			
SURFACE	DIRECTION OF LOADING	MAGNITUDE OF LOADING	CHORDWISE DISTRIBUTION
Horizontal Tail I	a) Up and Down	Figure A5 Curve (2)	See Figure A7
	b) Unsymmetrical Loading (Up and Down)	100% $\bar{w}$ on one side of airplane ℄ 65% $\bar{w}$ on other side of airplane ℄ for normal and utility categories. For acrobatic category see A23.11(c)	
Vertical Tail II	Right and Left	Figure A5 Curve (1)	Same as above
Aileron III	a) Up and Down	Figure A6 Curve (5)	(C) W ℄ Hinge
Wing Flap IV	a) Up	Figure A6 Curve (4)	(D) 2W W
	b) Down	.25 x Up Load (a)	
Trim Tab V	a) Up and Down	Figure A6 Curve (3)	Same as (D) above

NOTE: The surface loading I, II, III, and V above are based on speeds V_A min and V_C min.

The loading of IV is based on V_F min.

If values of speed greater than these minimums are selected for design, the appropriate surface loadings must be multiplied by the ratio $\left(\frac{V_{selected}}{V_{minimum}}\right)^2$.

For conditions I, II, III, and V the multiplying factor used must be the higher of

$$\left(\frac{V_A \text{ sel.}}{V_A \text{ min.}}\right)^2 \text{ or } \left(\frac{V_C \text{ sel.}}{V_C \text{ min.}}\right)^2$$

FIGURE A1—CHART FOR FINDING n_3 FACTOR AT SPEED V_C

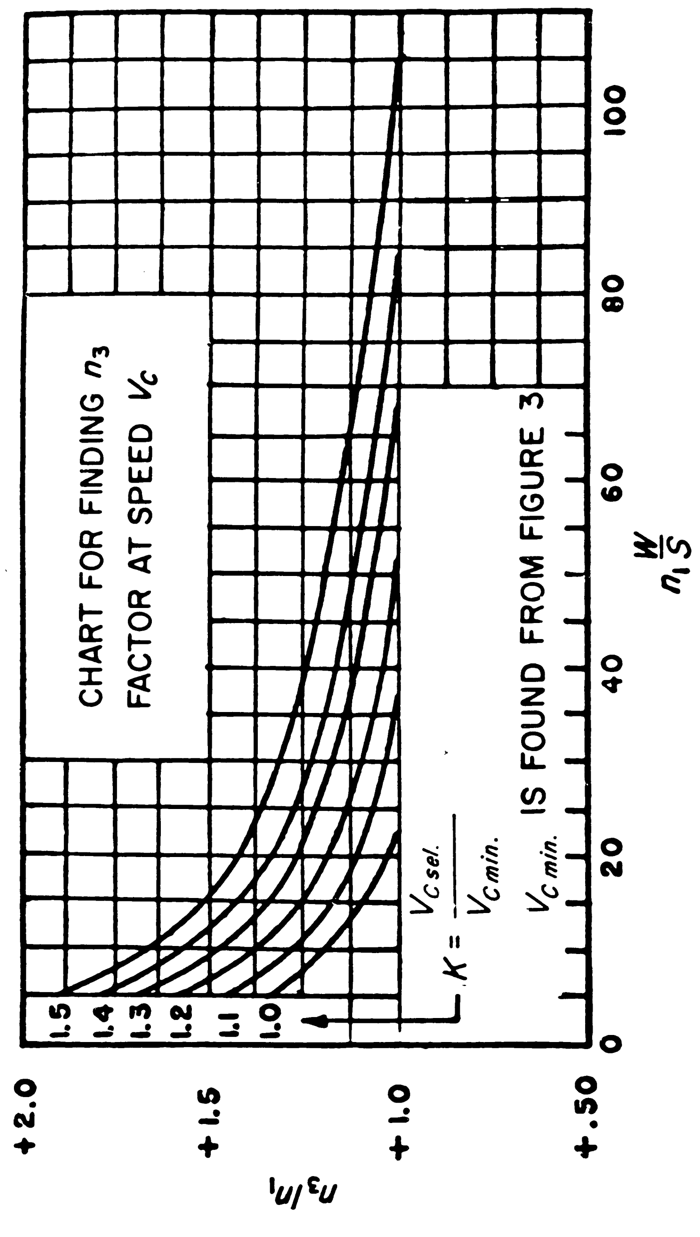


FIGURE A2 – CHART FOR FINDING n_4 FACTOR AT SPEED V_C

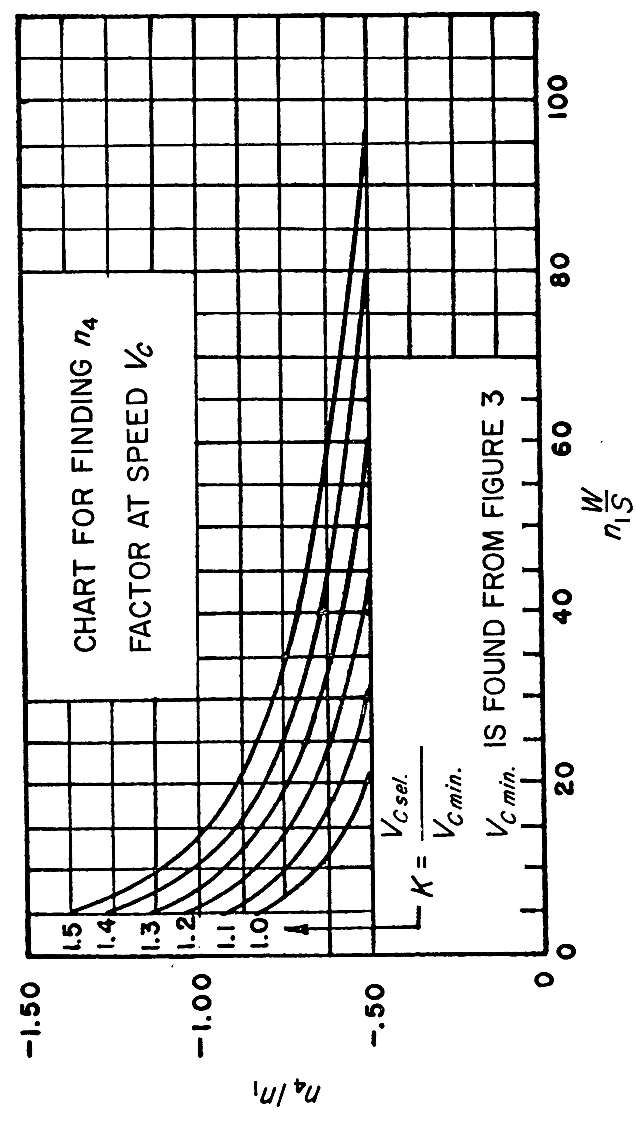
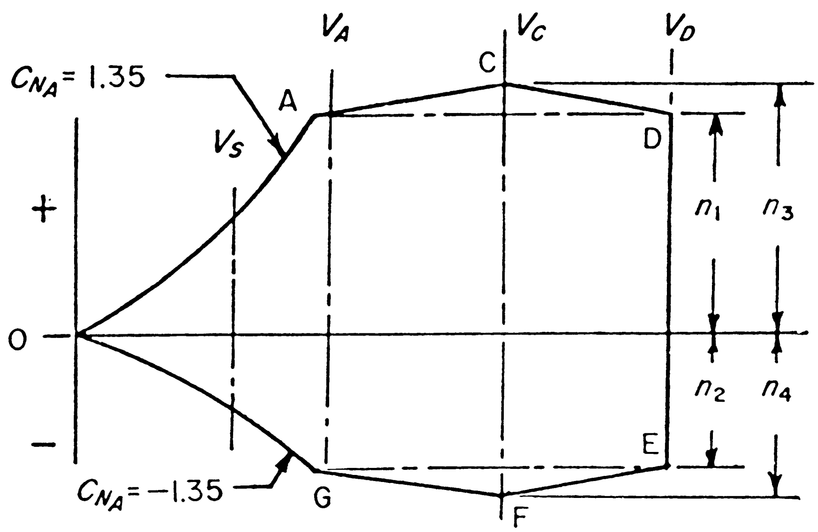


FIGURE A3—DETERMINATIONS OF MINIMUM DESIGN SPEEDS—EQUATIONS
SPEEDS ARE IN KNOTS

$V_{D\ min}=24.0\sqrt{n_1\frac{W}{S}}$ but need not exceed $\frac{1.4\sqrt{n_1}}{3.8}V_{C\ min}$;

$V_{C\ min}=17.0\sqrt{n_1\frac{W}{S}}$ but need not exceed 0.9 V_H; $V_{A\ min}=15.0\sqrt{n_1\frac{W}{S}}$ but need not exceed V_C used in design. $V_{F\ min}=11.0\sqrt{n_1\frac{W}{S}}$

FIGURE A-4—FLIGHT ENVELOPE



1. Conditions "C" or "F" need only be investigated when $n_3\frac{W}{S}$ or $n_4\frac{W}{S}$ is greater than $n_1\frac{W}{S}$ W |to , respectively.

2. Condition "G" need not be investigated when the supplementary condition specified in § 23.369 is investigated.

FIGURE A5 – AVERAGE LIMIT CONTROL SURFACE LOADING

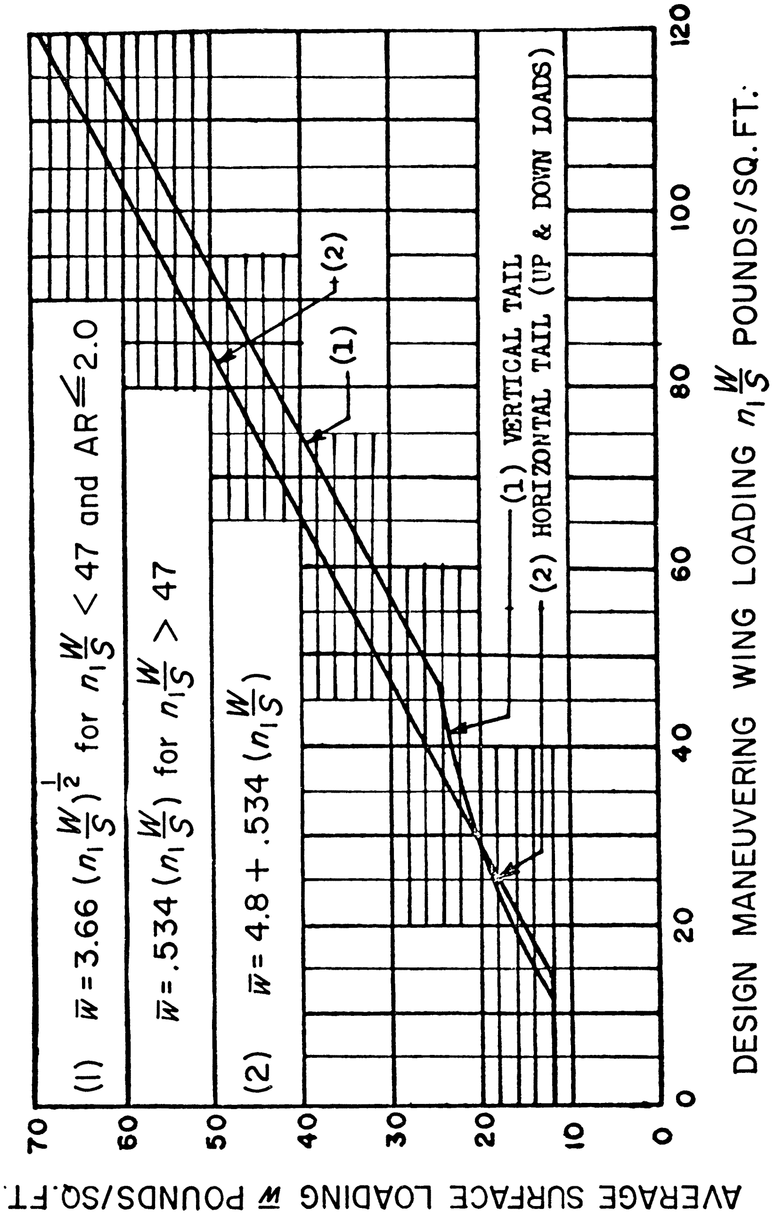


FIGURE A6 - AVERAGE LIMIT CONTROL SURFACE LOADING

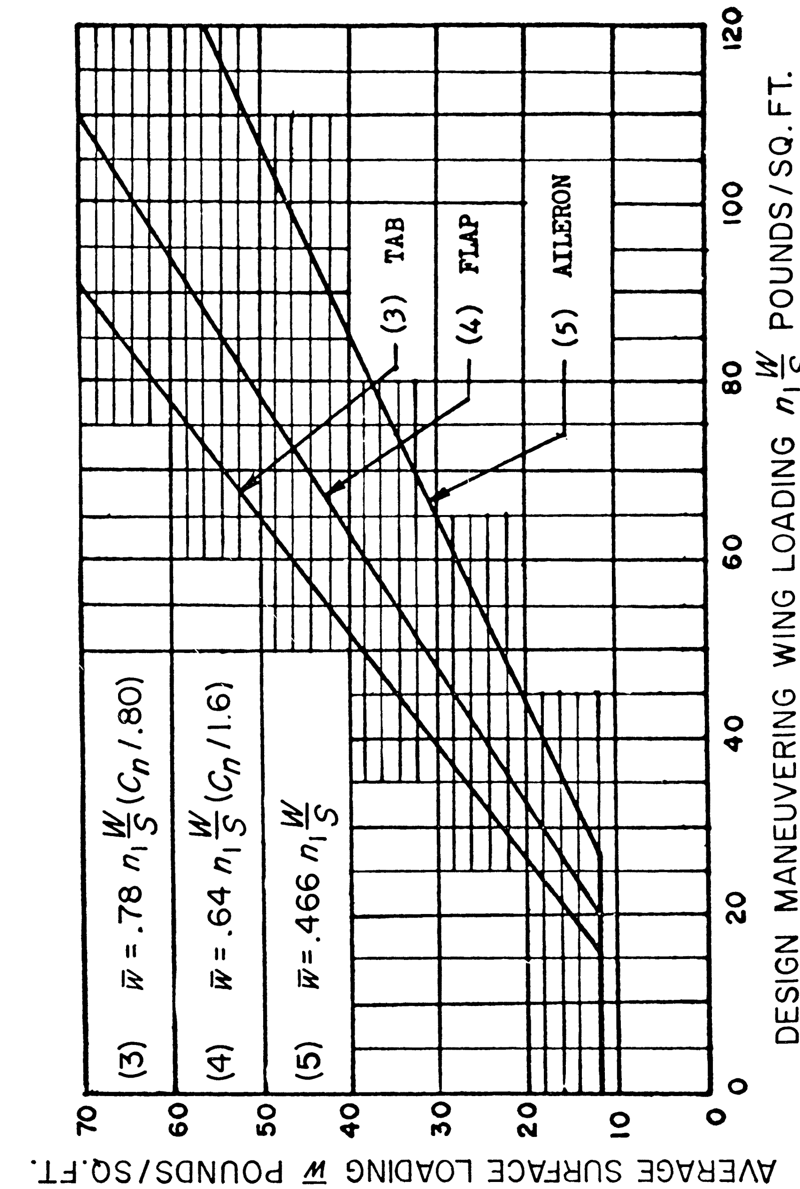
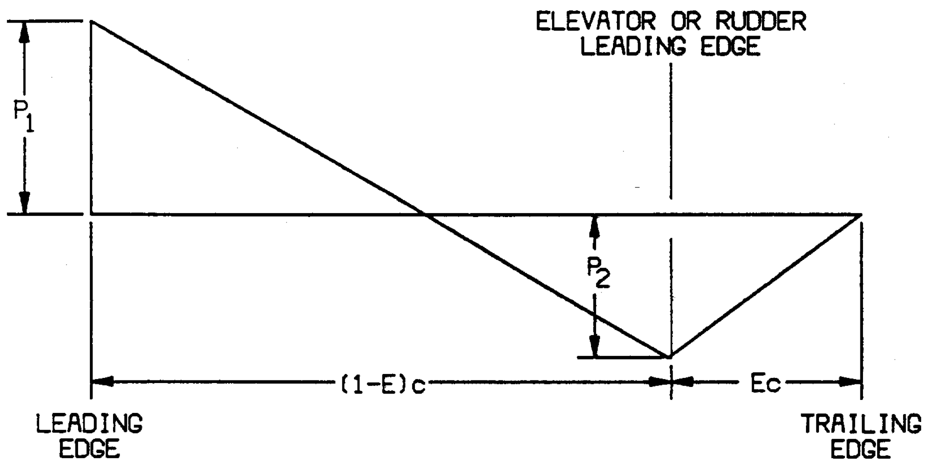


FIGURE A7—CHORDWISE LOAD DISTRIBUTION FOR STABILIZER AND ELEVATOR OR FIN AND RUDDER



$$P_1 = 2\,(\overline{w})\,\frac{(2 - E - 3d')}{(1 - E)}$$

$$P_2 = 2\,(\overline{w})\,(3d' + E - 1)$$

where:

$\bar{w}$=average surface loading (as specified in figure A.5)

E=ratio of elevator (or rudder) chord to total stabilizer and elevator (or fin and rudder) chord.

d′=ratio of distance of center of pressure of a unit spanwise length of combined stabilizer and elevator (or fin and rudder) measured from stabilizer (or fin) leading edge to the local chord. Sign convention is positive when center of pressure is behind leading edge.

c=local chord.

NOTE: Positive values of $\bar{w}$, P_1 and P_2 are all measured in the same direction.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23–7, 34 FR 13097, Aug. 13, 1969; 34 FR 14727, Sept. 24, 1969; Amdt. 23–16, 40 FR 2577, Jan. 14, 1975; Amdt. 23–28, 47 FR 13315, Mar. 29, 1982; Amdt. 23–48, 61 FR 5149, Feb. 9, 1996]

APPENDIX B TO PART 23 [RESERVED]

APPENDIX C TO PART 23—BASIC LANDING CONDITIONS

[C23.1 *Basic landing conditions*]

Condition	Tail wheel type		Nose wheel type		
	Level landing	Tail-down landing	Level landing with inclined reactions	Level landing with nose wheel just clear of ground	Tail-down landing
Reference section	23.479(a)(1)	23.481(a)(1)	23.479(a)(2)(i)	23.479(a)(2)(ii)	23.481(a)(2) and (b).
Vertical component at c. g	*nW*	*nW*	*nW*	*nW*	*nW.*
Fore and aft component at c. g	*KnW*	0	*KnW*	*KnW*	0.
Lateral component in either direction at c. g.	0	0	0	0	0.
Shock absorber extension (hydraulic shock absorber).	Note (2)	Note (2)	Note (2)	Note (2)	Note (2).
Shock absorber deflection (rubber or spring shock absorber), percent.	100	100	100	100	100.
Tire deflection	Static	Static	Static	Static	Static.
Main wheel loads (both wheels) (*Vr*)	(*n-L*)*W*	(*n-L*)*W* b/d	(*n-L*)*W* a′/d′	(*n-L*)*W*	(*n-L*)*W.*
Main wheel loads (both wheels) (*Dr*)	*KnW*	0	*KnW* a′/d′	*KnW*	0.

[C23.1 *Basic landing conditions*]

Condition	Tail wheel type		Nose wheel type		
	Level landing	Tail-down landing	Level landing with inclined reactions	Level landing with nose wheel just clear of ground	Tail-down landing
Tail (nose) wheel loads (*Vf*)	0	(*n-L*)*W* a/d	(*n-L*)*W* b′/d′	0	0.
Tail (nose) wheel loads (*Df*)	0	0	*KnW* b′/d′	0	0.
Notes	(1), (3), and (4).	(4)	(1)	(1), (3), and (4)	(3) and (4).

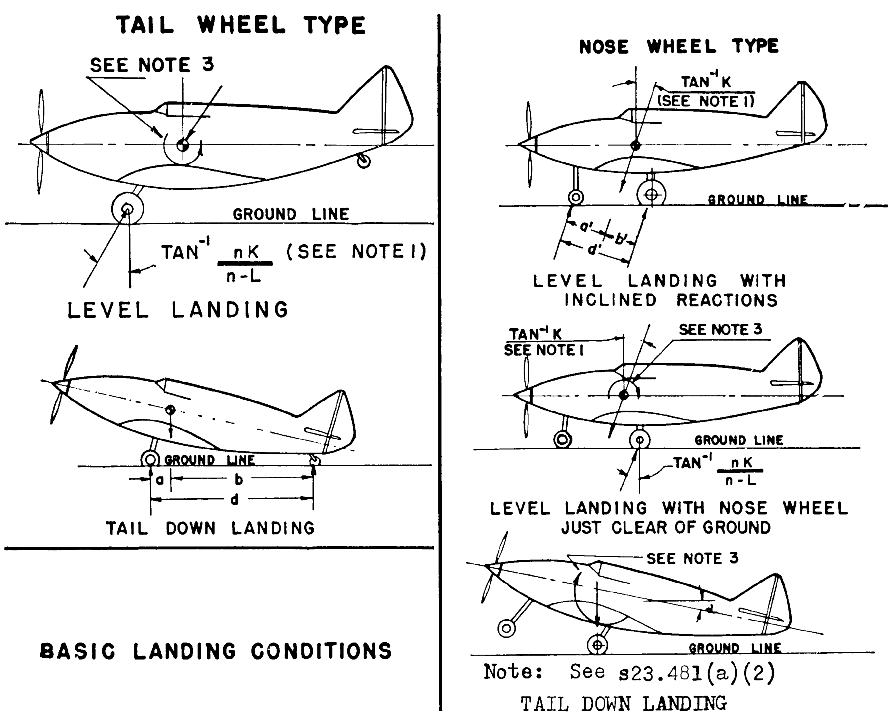
NOTE (1). *K* may be determined as follows: *K*=0.25 for *W*=3,000 pounds or less; *K*=0.33 for *W*=6,000 pounds or greater, with linear variation of *K* between these weights.

NOTE (2). For the purpose of design, the maximum load factor is assumed to occur throughout the shock absorber stroke from 25 percent deflection to 100 percent deflection unless otherwise shown and the load factor must be used with whatever shock absorber extension is most critical for each element of the landing gear.

NOTE (3). Unbalanced moments must be balanced by a rational or conservative method.

NOTE (4). *L* is defined in § 23.735(b).

NOTE (5). *n* is the limit inertia load factor, at the c.g. of the airplane, selected under § 23.473 (d), (f), and (g).



[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23–7, 34 FR 13099, Aug. 13, 1969]

APPENDIX D TO PART 23—WHEEL SPIN-UP AND SPRING-BACK LOADS

D23.1 *Wheel spin-up loads.*

(a) The following method for determining wheel spin-up loads for landing conditions is based on NACA T.N. 863. However, the drag component used for design may not be less than the drag load prescribed in § 23.479(b).

$F_{Hmax}=1/r_e \sqrt{2I_w(V_H-V_c)nF_{Vmax}/t_S}$

where—

F_{Hmax}=maximum rearward horizontal force acting on the wheel (in pounds);

r_e=effective rolling radius of wheel under impact based on recommended operating tire pressure (which may be assumed to be equal to the rolling radius under a static load of n_jW_e) in feet;

I_w=rotational mass moment of inertia of rolling assembly (in slug feet);

V_H=linear velocity of airplane parallel to ground at instant of contact (assumed to be 1.2 V_{S0}, in feet per second);

V_c=peripheral speed of tire, if prerotation is used (in feet per second) (there must be a positive means of pre-rotation before pre-rotation may be considered);

n=equals effective coefficient of friction (0.80 may be used);

F_{Vmax}=maximum vertical force on wheel (pounds)=n_jW_e, where W_e and n_j are defined in §23.725;

t_s=time interval between ground contact and attainment of maximum vertical force on wheel (seconds). (However, if the value of F_{Vmax}, from the above equation exceeds 0.8 F_{Vmax}, the latter value must be used for F_{Hmax}.)

(b) The equation assumes a linear variation of load factor with time until the peak load is reached and under this assumption, the equation determines the drag force at the time that the wheel peripheral velocity at radius r_e equals the airplane velocity. Most shock absorbers do not exactly follow a linear variation of load factor with time. Therefore, rational or conservative allowances must be made to compensate for these variations. On most landing gears, the time for wheel spin-up will be less than the time required to develop maximum vertical load factor for the specified rate of descent and forward velocity. For exceptionally large wheels, a wheel peripheral velocity equal to the ground speed may not have been attained at the time of maximum vertical gear load. However, as stated above, the drag spin-up load need not exceed 0.8 of the maximum vertical loads.

(c) Dynamic spring-back of the landing gear and adjacent structure at the instant just after the wheels come up to speed may result in dynamic forward acting loads of considerable magnitude. This effect must be determined, in the level landing condition, by assuming that the wheel spin-up loads calculated by the methods of this appendix are reversed. Dynamic spring-back is likely to become critical for landing gear units having wheels of large mass or high landing speeds.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23–45, 58 FR 42167, Aug. 6, 1993]

APPENDIX E TO PART 23 [RESERVED]

APPENDIX F TO PART 23—TEST PROCEDURE

Acceptable test procedure for self-extinguishing materials for showing compliance with §§23.853, 23.855 and 23.1359.

(a) *Conditioning.* Specimens must be conditioned to 70 degrees F, plus or minus 5 degrees, and at 50 percent plus or minus 5 percent relative humidity until moisture equilibrium is reached or for 24 hours. Only one specimen at a time may be removed from the conditioning environment immediately before subjecting it to the flame.

(b) *Specimen configuration.* Except as provided for materials used in electrical wire and cable insulation and in small parts, materials must be tested either as a section cut from a fabricated part as installed in the airplane or as a specimen simulating a cut section, such as: a specimen cut from a flat sheet of the material or a model of the fabricated part. The specimen may be cut from any location in a fabricated part; however, fabricated units, such as sandwich panels, may not be separated for a test. The specimen thickness must be no thicker than the minimum thickness to be qualified for use in the airplane, except that: (1) Thick foam parts, such as seat cushions, must be tested in ½ inch thickness; (2) when showing compliance with §23.853(d)(3)(v) for materials used in small parts that must be tested, the materials must be tested in no more than ⅛ inch thickness; (3) when showing compliance with §23.1359(c) for materials used in electrical wire and cable insulation, the wire and cable specimens must be the same size as used in the airplane. In the case of fabrics, both the warp and fill direction of the weave must be tested to determine the most critical flammability conditions. When performing the tests prescribed in paragraphs (d) and (e) of this appendix, the specimen must be mounted in a metal frame so that (1) in the vertical tests of paragraph (d) of this appendix, the two long edges and the upper edge are held securely; (2) in the horizontal test of paragraph (e) of this appendix, the two long edges and the edge away from the flame are held securely; (3) the exposed area of the specimen is at least 2 inches wide and 12 inches long, unless the actual size used in the airplane is smaller; and (4) the edge to which the burner flame is applied must not consist of the finished or protected edge of the specimen but must be representative of the actual cross section of the material or part installed in the airplane. When performing the test prescribed in paragraph (f) of this appendix, the specimen must be mounted in metal frame so that all four edges are held securely and the exposed area of the specimen is at least 8 inches by 8 inches.

(c) *Apparatus.* Except as provided in paragraph (g) of this appendix, tests must be conducted in a draft-free cabinet in accordance with Federal Test Method Standard 191 Method 5903 (revised Method 5902) which is available from the General Services Administration, Business Service Center, Region 3, Seventh and D Streets SW., Washington, D.C. 20407, or with some other approved equivalent method. Specimens which are too large for the cabinet must be tested in similar draft-free conditions.

(d) *Vertical test.* A minimum of three specimens must be tested and the results averaged. For fabrics, the direction of weave corresponding to the most critical flammability conditions must be parallel to the longest dimension. Each specimen must be supported vertically. The specimen must be exposed to a Bunsen or Tirrill burner with a nominal ⅜-inch I.D. tube adjusted to give a flame of 1½ inches in height. The minimum flame temperature measured by a calibrated thermocouple pryometer in the center of the flame must be 1550 °F. The lower edge of the specimen must be three-fourths inch above the top edge of the burner. The flame must be applied to the center line of the lower edge of the specimen. For materials covered by §§23.853(d)(3)(i) and 23.853(f), the flame must be applied for 60 seconds and then removed. For materials covered by §23.853(d)(3)(ii), the flame must be applied for 12 seconds and then removed. Flame time, burn length, and flaming time of drippings, if any, must be recorded. The burn length determined in accordance with paragraph (h) of this appendix must be measured to the nearest one-tenth inch.

(e) *Horizontal test.* A minimum of three specimens must be tested and the results averaged. Each specimen must be supported horizontally. The exposed surface when installed in the airplane must be face down for the test. The specimen must be exposed to a Bunsen burner or Tirrill burner with a nominal ⅜-inch I.D. tube adjusted to give a flame of 1½ inches in height. The minimum flame temperature measured by a calibrated thermocouple pyrometer in the center of the flame must be 1550 °F. The specimen must be positioned so that the edge being tested is three-fourths of an inch above the top of, and on the center line of, the burner. The flame must be applied for 15 seconds and then removed. A minimum of 10 inches of the specimen must be used for timing purposes, approximately 1½ inches must burn before the burning front reaches the timing zone, and the average burn rate must be recorded.

(f) *Forty-five degree test.* A minimum of three specimens must be tested and the results averaged. The specimens must be supported at an angle of 45 degrees to a horizontal surface. The exposed surface when installed in the aircraft must be face down for the test. The specimens must be exposed to a Bunsen or Tirrill burner with a nominal ⅜ inch I.D. tube adjusted to give a flame of 1½ inches in height. The minimum flame temperature measured by a calibrated thermocouple pyrometer in the center of the flame must be 1550 °F. Suitable precautions must be taken to avoid drafts. The flame must be applied for 30 seconds with one-third contacting the material at the center of the specimen and then removed. Flame time, glow time, and whether the flame penetrates (passes through) the specimen must be recorded.

(g) *Sixty-degree test.* A minimum of three specimens of each wire specification (make and size) must be tested. The specimen of wire or cable (including insulation) must be placed at an angle of 60 degrees with the horizontal in the cabinet specified in paragraph (c) of this appendix, with the cabinet door open during the test or placed within a chamber approximately 2 feet high × 1 foot × 1 foot, open at the top and at one vertical side (front), that allows sufficient flow of air for complete combustion but is free from drafts. The specimen must be parallel to and approximately 6 inches from the front of the chamber. The lower end of the specimen must be held rigidly clamped. The upper end of the specimen must pass over a pulley or rod and must have an appropriate weight attached to it so that the specimen is held tautly throughout the flammability test. The test specimen span between lower clamp and upper pulley or rod must be 24 inches and must be marked 8 inches from the lower end to indicate the central point for flame application. A flame from a Bunsen or Tirrill burner must be applied for 30 seconds at the test mark. The burner must be mounted underneath the test mark on the specimen, perpendicular to the specimen and at an angle of 30 degrees to the vertical plane of the specimen. The burner must have a nominal bore of three-eighths inch, and must be adjusted to provide a three-inch-high flame with an inner cone approximately one-third of the flame height. The minimum temperature of the hottest portion of the flame, as measured with a calibrated thermocouple pyrometer, may not be less than 1,750 °F. The burner must be positioned so that the hottest portion of the flame is applied to the test mark on the wire. Flame time, burn length, and flaming time drippings, if any, must be recorded. The burn length determined in accordance with paragraph (h) of this appendix must be measured to the nearest one-tenth inch. Breaking of the wire specimen is not considered a failure.

(h) *Burn length.* Burn length is the distance from the original edge to the farthest evidence of damage to the test specimen due to flame impingement, including areas of partial or complete consumption, charring, or embrittlement, but not including areas sooted, stained, warped, or discolored, nor areas where material has shrunk or melted away from the heat source.

[Amdt. 23–23, 43 FR 50594, Oct. 30, 1978, as amended by Amdt. 23–34, 52 FR 1835, Jan. 15, 1987; 52 FR 34745, Sept. 14, 1987; Amdt. 23–49, 61 FR 5170, Feb. 9, 1996]

APPENDIX G TO PART 23—INSTRUCTIONS FOR CONTINUED AIRWORTHINESS

G23.1 *General.* (a) This appendix specifies requirements for the preparation of Instructions for Continued Airworthiness as required by §23.1529.

(b) The Instructions for Continued Airworthiness for each airplane must include the Instructions for Continued Airworthiness for each engine and propeller (hereinafter designated 'products'), for each appliance required by this chapter, and any required information relating to the interface of those appliances and products with the airplane. If Instructions for Continued Airworthiness are not supplied by the manufacturer of an appliance or product installed in the airplane, the Instructions for Continued Airworthiness for the airplane must include the information essential to the continued airworthiness of the airplane.

(c) The applicant must submit to the FAA a program to show how changes to the Instructions for Continued Airworthiness made by the applicant or by the manufacturers of products and appliances installed in the airplane will be distributed.

G23.2 *Format.* (a) The Instructions for Continued Airworthiness must be in the form of a manual or manuals as appropriate for the quantity of data to be provided.

(b) The format of the manual or manuals must provide for a practical arrangement.

G23.3 *Content.* The contents of the manual or manuals must be prepared in the English language. The Instructions for Continued Airworthiness must contain the following manuals or sections, as appropriate, and information:

(a) *Airplane maintenance manual or section.* (1) Introduction information that includes an explanation of the airplane's features and data to the extent necessary for maintenance or preventive maintenance.

(2) A description of the airplane and its systems and installations including its engines, propellers, and appliances.

(3) Basic control and operation information describing how the airplane components and systems are controlled and how they operate, including any special procedures and limitations that apply.

(4) Servicing information that covers details regarding servicing points, capacities of tanks, reservoirs, types of fluids to be used, pressures applicable to the various systems, location of access panels for inspection and servicing, locations of lubrication points, lubricants to be used, equipment required for servicing, tow instructions and limitations, mooring, jacking, and leveling information.

(b) *Maintenance instructions.* (1) Scheduling information for each part of the airplane and its engines, auxiliary power units, propellers, accessories, instruments, and equipment that provides the recommended periods at which they should be cleaned, inspected, adjusted, tested, and lubricated, and the degree of inspection, the applicable wear tolerances, and work recommended at these periods. However, the applicant may refer to an accessory, instrument, or equipment manufacturer as the source of this information if the applicant shows that the item has an exceptionally high degree of complexity requiring specialized maintenance techniques, test equipment, or expertise. The recommended overhaul periods and necessary cross reference to the Airworthiness Limitations section of the manual must also be included. In addition, the applicant must include an inspection program that includes the frequency and extent of the inspections necessary to provide for the continued airworthiness of the airplane.

(2) Troubleshooting information describing probable malfunctions, how to recognize those malfunctions, and the remedial action for those malfunctions.

(3) Information describing the order and method of removing and replacing products and parts with any necessary precautions to be taken.

(4) Other general procedural instructions including procedures for system testing during ground running, symmetry checks, weighing and determining the center of gravity, lifting and shoring, and storage limitations.

(c) Diagrams of structural access plates and information needed to gain access for inspections when access plates are not provided.

(d) Details for the application of special inspection techniques including radiographic and ultrasonic testing where such processes are specified.

(e) Information needed to apply protective treatments to the structure after inspection.

(f) All data relative to structural fasteners such as identification, discard recommendations, and torque values.

(g) A list of special tools needed.

(h) In addition, for commuter category airplanes, the following information must be furnished:

(1) Electrical loads applicable to the various systems;

(2) Methods of balancing control surfaces;

(3) Identification of primary and secondary structures; and

(4) Special repair methods applicable to the airplane.

G23.4 *Airworthiness Limitations section.* The Instructions for Continued Airworthiness must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section must set forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure required for type certification. If the Instructions for Continued Airworthiness consist of multiple documents, the section required by this paragraph must be included in the principal manual. This section must contain a legible statement in a prominent location that reads: "The Airworthiness Limitations section is FAA approved and specifies maintenance required under §§43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved."

[Amdt. 23–26, 45 FR 60171, Sept. 11, 1980, as amended by Amdt. 23–34, 52 FR 1835, Jan. 15, 1987; 52 FR 34745, Sept. 14, 1987; Amdt. 23–37, 54 FR 34329, Aug. 18, 1989]

Appendix H to Part 23—Installation of An Automatic Power Reserve (APR) System

H23.1, *General.*

(a) This appendix specifies requirements for installation of an APR engine power control system that automatically advances power or thrust on the operating engine(s) in the event any engine fails during takeoff.

(b) With the APR system and associated systems functioning normally, all applicable requirements (except as provided in this appendix) must be met without requiring any action by the crew to increase power or thrust.

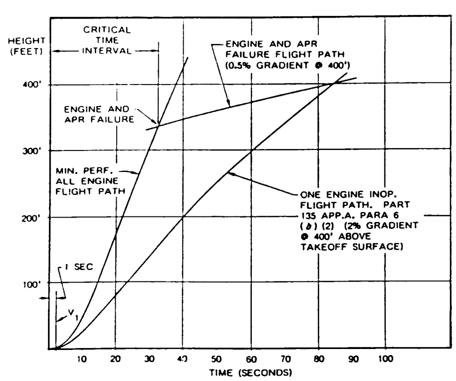
H23.2, *Definitions.*

(a) *Automatic power reserve system* means the entire automatic system used only during takeoff, including all devices both mechanical and electrical that sense engine failure, transmit signals, actuate fuel controls or power levers on operating engines, including power sources, to achieve the scheduled power increase and furnish cockpit information on system operation.

(b) *Selected takeoff power*, notwithstanding the definition of "Takeoff Power" in part 1 of the Federal Aviation Regulations, means the power obtained from each initial power setting approved for takeoff.

(c) *Critical Time Interval*, as illustrated in figure H1, means that period starting at V_1 minus one second and ending at the intersection of the engine and APR failure flight path line with the minimum performance all engine flight path line. The engine and APR failure flight path line intersects the one-engine-inoperative flight path line at 400 feet above the takeoff surface. The engine and APR failure flight path is based on the airplane's performance and must have a positive gradient of at least 0.5 percent at 400 feet above the takeoff surface.

FIGURE H1 - CRITICAL TIME INTERVAL ILLUSTRATION.



H23.3, *Reliability and performance requirements.*

(a) It must be shown that, during the critical time interval, an APR failure that increases or does not affect power on either engine will not create a hazard to the airplane, or it must be shown that such failures are improbable.

(b) It must be shown that, during the critical time interval, there are no failure modes of the APR system that would result in a failure that will decrease the power on either engine or it must be shown that such failures are extremely improbable.

(c) It must be shown that, during the critical time interval, there will be no failure of the APR system in combination with an engine failure or it must be shown that such failures are extremely improbable.

(d) All applicable performance requirements must be met with an engine failure occurring at the most critical point during takeoff with the APR system functioning normally.

H23.4, *Power setting*.

The selected takeoff power set on each engine at the beginning of the takeoff roll may not be less than—

(a) The power necessary to attain, at V_1, 90 percent of the maximum takeoff power approved for the airplane for the existing conditions;

(b) That required to permit normal operation of all safety-related systems and equipment that are dependent upon engine power or power lever position; and

(c) That shown to be free of hazardous engine response characteristics when power is advanced from the selected takeoff power level to the maximum approved takeoff power.

H23.5, *Powerplant controls—general*.

(a) In addition to the requirements of §23.1141, no single failure or malfunction (or probable combination thereof) of the APR, including associated systems, may cause the failure of any powerplant function necessary for safety.

(b) The APR must be designed to—

(1) Provide a means to verify to the flight crew before takeoff that the APR is in an operating condition to perform its intended function;

(2) Automatically advance power on the operating engines following an engine failure during takeoff to achieve the maximum attainable takeoff power without exceeding engine operating limits;

(3) Prevent deactivation of the APR by manual adjustment of the power levers following an engine failure;

(4) Provide a means for the flight crew to deactivate the automatic function. This means must be designed to prevent inadvertent deactivation; and

(5) Allow normal manual decrease or increase in power up to the maximum takeoff power approved for the airplane under the existing conditions through the use of power levers, as stated in §23.1141(c), except as provided under paragraph (c) of H23.5 of this appendix.

(c) For airplanes equipped with limiters that automatically prevent engine operating limits from being exceeded, other means may be used to increase the maximum level of power controlled by the power levers in the event of an APR failure. The means must be located on or forward of the power levers, must be easily identified and operated under all operating conditions by a single action of any pilot with the hand that is normally used to actuate the power levers, and must meet the requirements of §23.777 (a), (b), and (c).

H23.6, *Powerplant instruments*.

In addition to the requirements of §23.1305:

(a) A means must be provided to indicate when the APR is in the armed or ready condition.

(b) If the inherent flight characteristics of the airplane do not provide warning that an engine has failed, a warning system independent of the APR must be provided to give the pilot a clear warning of any engine failure during takeoff.

(c) Following an engine failure at V_1 or above, there must be means for the crew to readily and quickly verify that the APR has operated satisfactorily.

[Doc. 26344, 58 FR 18979, Apr. 9, 1993]

APPENDIX I TO PART 23—SEAPLANE LOADS

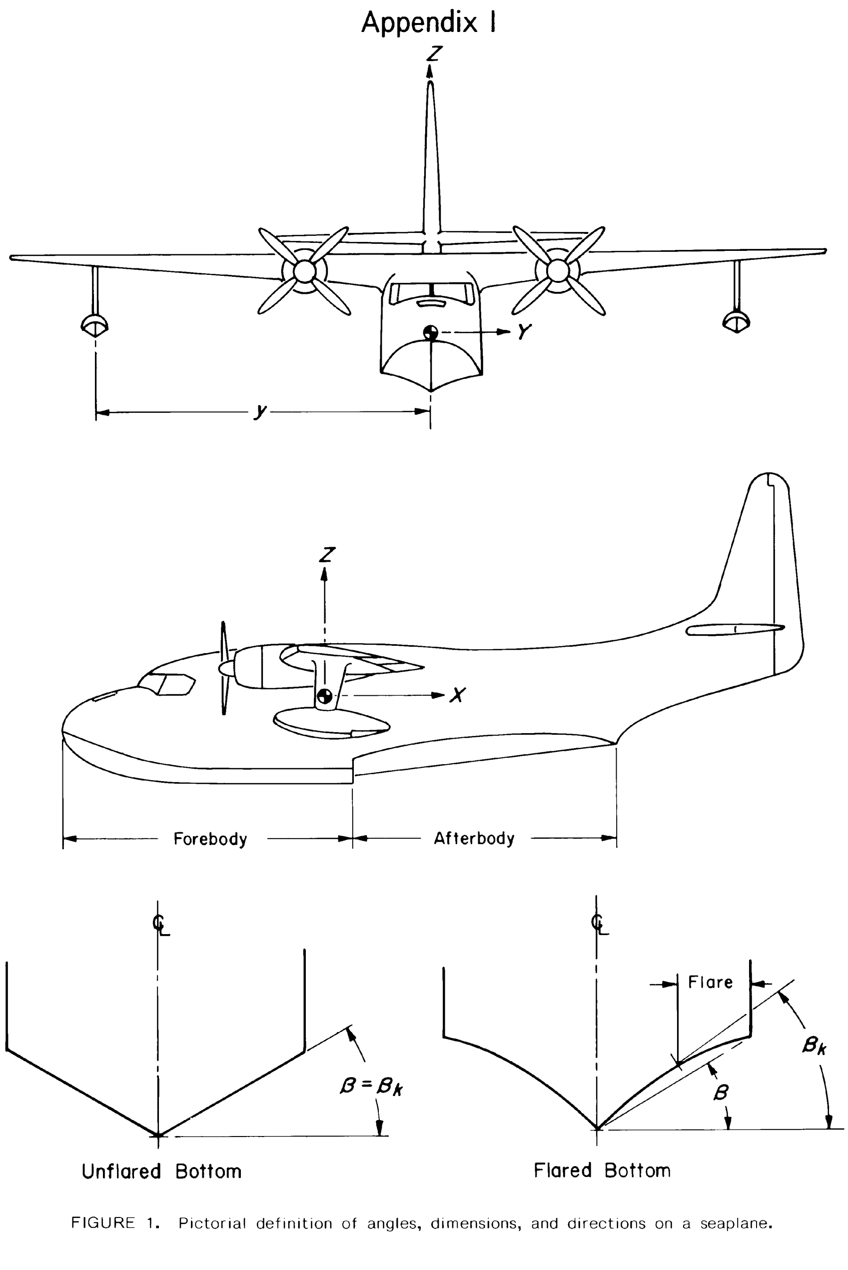


FIGURE 1. Pictorial definition of angles, dimensions, and directions on a seaplane.

Appendix I (continued)

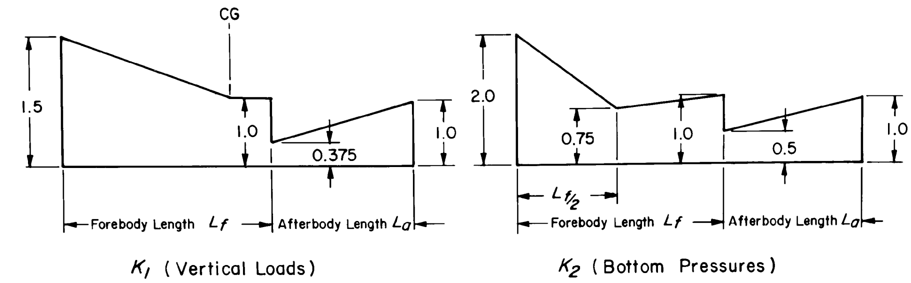


FIGURE 2. Hull station weighing factor.

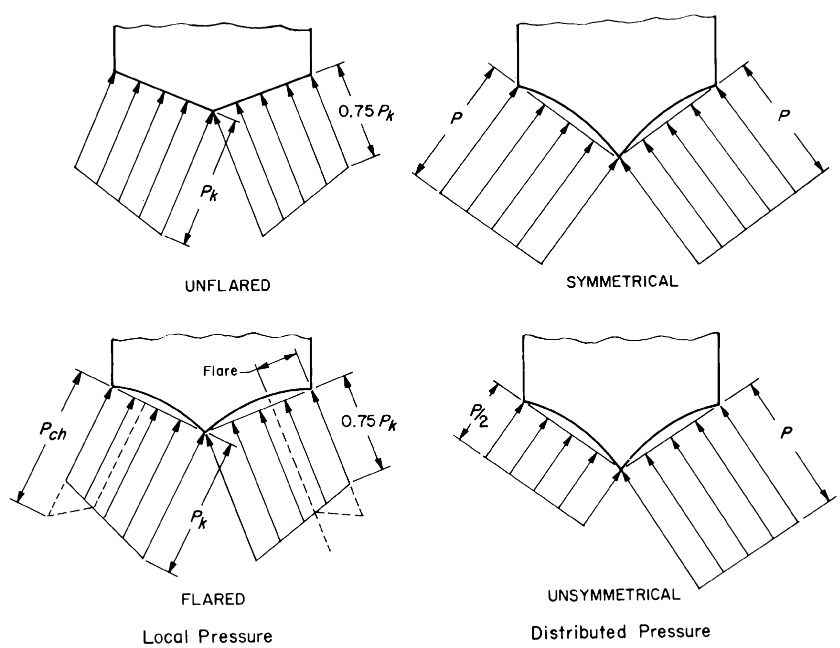


FIGURE 3. Transverse pressure distributions.

[Amdt. 23–45, 58 FR 42167, Aug. 6, 1993; 58 FR 51970, Oct. 5, 1993]

APPENDIX J TO PART 23—HIRF ENVIRONMENTS AND EQUIPMENT HIRF TEST LEVELS

This appendix specifies the HIRF environments and equipment HIRF test levels for electrical and electronic systems under §23.1308. The field strength values for the HIRF environments and equipment HIRF test levels are expressed in root-mean-square units measured during the peak of the modulation cycle.

(a) HIRF environment I is specified in the following table:

TABLE I.—HIRF ENVIRONMENT I

Frequency	Field strength (volts/meter)	
	Peak	Average
10 kHz–2 MHz	50	50
2 MHz–30 MHz	100	100
30 MHz–100 MHz	50	50
100 MHz–400 MHz	100	100
400 MHz–700 MHz	700	50
700 MHz–1 GHz	700	100
GHz–2 GHz	2,000	200
2 GHz–6 GHz	3,000	200
6 GHz–8 GHz	1,000	200
8 GHz–12 GHz	3,000	300
12 GHz–18 GHz	2,000	200
18 GHz–40 GHz	600	200

In this table, the higher field strength applies at the frequency band edges.

(b) HIRF environment II is specified in the following table:

TABLE II.–HIRF ENVIRONMENT II

Frequency	Field strength (volts/meter)	
	Peak	Average
10 kHz–500 kHz	20	20
500 kHz–2 MHz	30	30
2 MHz–30 MHz	100	100
30 MHz–100 MHz	10	10
100 MHz–200 MHz	30	10
200 MHz–400 MHz	10	10
400 MHz–1 GHz	700	40
1 GHz–2 GHz	1,300	160
2 GHz–4 GHz	3,000	120
4 GHz–6 GHz	3,000	160
6 GHz–8 GHz	400	170
8 GHz–12 GHz	1,230	230
12 GHz–18 GHz	730	190
18 GHz–40 GHz	600	150

In this table, the higher field strength applies at the frequency band edges.

(c) *Equipment HIRF Test Level 1.*

(1) From 10 kilohertz (kHz) to 400 megahertz (MHz), use conducted susceptibility tests with continuous wave (CW) and 1 kHz square wave modulation with 90 percent depth or greater. The conducted susceptibility current must start at a minimum of 0.6 milliamperes (mA) at 10 kHz, increasing 20 decibels (dB) per frequency decade to a minimum of 30 mA at 500 kHz.

(2) From 500 kHz to 40 MHz, the conducted susceptibility current must be at least 30 mA.

(3) From 40 MHz to 400 MHz, use conducted susceptibility tests, starting at a minimum of 30 mA at 40 MHz, decreasing 20 dB per frequency decade to a minimum of 3 mA at 400 MHz.

(4) From 100 MHz to 400 MHz, use radiated susceptibility tests at a minimum of 20 volts per meter (V/m) peak with CW and 1 kHz square wave modulation with 90 percent depth or greater.

(5) From 400 MHz to 8 gigahertz (GHz), use radiated susceptibility tests at a minimum of 150 V/m peak with pulse modulation of 4 percent duty cycle with a 1 kHz pulse repetition frequency. This signal must be switched on and off at a rate of 1 Hz with a duty cycle of 50 percent.

(d) *Equipment HIRF Test Level 2.* Equipment HIRF test level 2 is HIRF environment II in table II of this appendix reduced by acceptable aircraft transfer function and attenuation curves. Testing must cover the frequency band of 10 kHz to 8 GHz.

(e) *Equipment HIRF Test Level 3.*

(1) From 10 kHz to 400 MHz, use conducted susceptibility tests, starting at a minimum of 0.15 mA at 10 kHz, increasing 20 dB per frequency decade to a minimum of 7.5 mA at 500 kHz.

(2) From 500 kHz to 40 MHz, use conducted susceptibility tests at a minimum of 7.5 mA.

(3) From 40 MHz to 400 MHz, use conducted susceptibility tests, starting at a minimum of 7.5 mA at 40 MHz, decreasing 20 dB per frequency decade to a minimum of 0.75 mA at 400 MHz.

(4) From 100 MHz to 8 GHz, use radiated susceptibility tests at a minimum of 5 V/m.

[Doc. No. FAA–2006–23657, 72 FR 44025, Aug. 6, 2007]

PART 25—AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY AIRPLANES

SPECIAL FEDERAL AVIATION REGULATION NO. 13

Subpart A—General

Sec.

Subpart B—Flight

GENERAL

PERFORMANCE